Pleiades High Resolution Satellite :
a Solution for Military and Civilian Needs in Metric-Class Optical Observation

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Abstract

In 2000, Cnes (French space agency) decided to initiate the Pleiades program consisting in a constellation of European satellite components dedicated to Earth observation. Astrium is in charge of the first French component, named Pleiades High Resolution, as satellite prime contractor. This satellite aims at fulfilling the needs of French and Italian governments, for both civilian and military users, in metric-class optical observation at the 2006 horizon.

This small satellite weighting around 900 kg will orbit at 695 km on a Sun synchronous orbit. It will provide colored images of more than 20 km swath, with a panchromatic resolution better than 1 meter, and a multi-spectral resolution of 2.8 m at nadir. The image data handling chain will make maximum use of the onboard 600 Gbits solid state mass memory, and of the downlink at 600 Mbps in X-band. The satellite will provide high image quality performances, characterized by a highly stable dynamic behavior, and an autonomous image location better than 20 m (maximum). This highly agile spacecraft will maneuver 60 deg in less than 25 sec. To support those performances, new technologies will be used, such as light material with high thermal stability for the instrument structure, integrated detection electronics near the focal plane, Control Moment Gyros for agility, Fiber Optic Gyros for high accuracy attitude measurement, Li-ion batteries and triple junction solar cells for a power system maximum efficiency.

After a short mission introduction, the paper presents the satellite design with its different functional chains. The satellite performances are then presented.

Program Context

Program Context

Earth observation has always been a priority for French space policy. In this frame, Cnes decided to start in early 2000 the Pleiades program, defined as a multi-sensor constellation dedicated to European needs in remote sensing.

The constellation first two components are the Italian radar satellites (Cosmo-Skymed program) and the French optical observation satellites, called Pleiades. Those satellites will be developed in the same time frame, and their image products will be shared by the French and Italian agencies. This dual system will be used for both the civilian and the military needs in France and Italy.

The French 'Pleiades High Resolution' is scheduled for launch in 2006, and will provide metric-class colored images. Therefore, Cnes decided to initiate a feasibility study in 2000, completed successfully in 2001. The present paper presents the main study results, and the Pleiades satellite architecture.

The hereafter described satellite takes full benefit from the existing developments at Astrium on the Leostar platform, and from the optical observation background, namely the Spot and Helios family. Nevertheless, this satellite is considered as the first of the next French observation generation, because of its high performances, and of the design solutions implemented.

Main Requirements

Mission Highlights

The mission consists in Earth optical observation in panchromatic band of 0.7 m resolution at nadir, and in four multi-spectral band of 2.8 m resolution at nadir. The image swath shall be larger than 20 km.

The Pleiades satellite will orbit the Earth during 5 years on a quasi-circular, Sun-synchronous orbit of 695 km altitude, at a local hour at descending node of 10 h. The satellite shall be designed to maneuver up to 50° from nadir, and to keep its nominal specified performances up to 30° from nadir. It shall reach a maneuver angle of 5° in less than 6 s, and 60° in less than 25 s.

The image storage capacity shall reach 600 Gbits at end of life, while the image downlink rate shall be 600 Mbits/s.

Image Quality

High image quality is required. The panchromatic Modulation Transfer Function (MTF) is of 0.09, while the Signal to Noise Ratio (SNR) at nominal Earth radiance is of 90. Images shall be located without external references to better than 20 m (max.), and with ground control points spaced by 80 km to better than 0.5 m (max.).

Satellite Design

Satellite

The satellite is physically divided in two parts : the instrument realizing the imaging function by a telescope, a detection unit and their proximity electronics, and the bus, gathering the rest of the satellite functions, namely payload data handling and transmission, mechanical support, thermal control, power generation and distribution, attitude control, on-board data handling. The design solutions for Pleiades are presented for the satellite (mechanical and thermal overall concept), the instrument, and then the bus.

Mechanical Architecture

The main design drivers for the satellite architecture are the agility and the image location accuracy. A high agility requires a very compact design, with a few stiff appendages. As a consequence, the instrument is integrated inside the bus (with some margins in case of a potential instrument focal length increase). A high image location accuracy is achieved by minimizing the interface between the instrument and the bus : the instrument is directly supported by the launcher interface cone inside the bus, providing a controlled thermal environment and a limited interface deformation variation. The three star trackers and the gyroscope heads are directly supported by the instrument to avoid any thermal distortion that could be induced by the bus.

The bus structure is build on an hexagonal shape, with three solar arrays at 120 deg, and three star trackers in a quasi tetrahedron configuration, optimizing the attitude determination accuracy. This configuration authorizes an easy accommodation of the instrument focal plane radiator for maximum heat dissipation. An antenna support structure is used to carry the Earth-pointing antennas and for the instrument baffle.
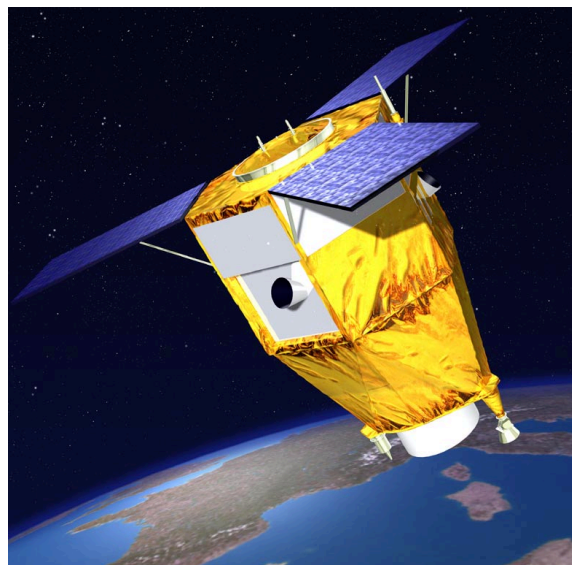
Figure 1 : Satellite In-Flight Configuration

The solar arrays are mounted directly on the bus structure without any drive mechanism to ensure a maximum stability. The array size is minimized by using high-efficiency triple junction solar cells. Their

stiffness is increased by the use of Carpentier joints when deployed.

Thermal Control

One of the main advantages of the Pleiades configuration is to provide a controlled thermal environment to the instrument, which simplifies its own thermal control to paintings, heaters, and wrapped Multi Layer Insulation (MLI).

This is achieved by maintaining a bus temperature around 10°C, cooler than the instrument controlled at about 20°C. The bus temperature itself is controlled through the lateral panels radiators looking towards cold space, with local heaters used for dedicated units like the battery.

The focal plane assembly has a dedicated thermal control unit using heat pipes to keep a very stable temperature, as it is separated from the radiator panel by the proximity electronics.

The star trackers have their own radiative thermal control to avoid any coupling with the instrument.

Electrical Architecture

The satellite electrical architecture is organized around a central computer, based on a SPARC ERC 32 computer, communicating via 1553 buses to the onboard equipment, as shown in Figure 2.
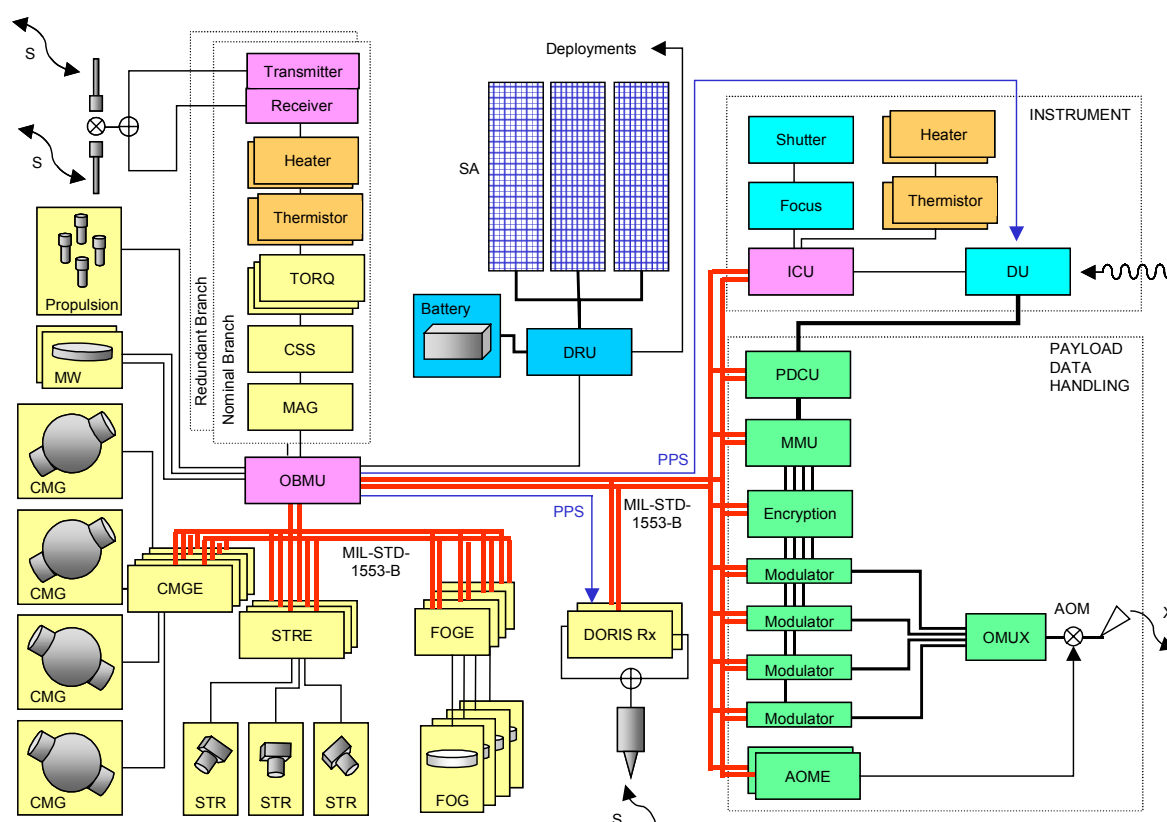
Figure 2 : Satellite Electrical Architecture

The computing, monitoring and reconfiguration functions are centralized in the On Board Management Unit (OBMU). The bus equipment interfaces are gathered in the same unit, which includes the analog sensors and actuators interface, the propulsion interface and the bus thermal control hardware interface. The Instrument Management Unit (IMU) gathers the instrument interfaces : instrument thermal control hardware, mechanisms command, detection unit power conversion. All the other equipments have their own 1553 interface, and directly interface the OBMU. Two 1553 buses are nominally used : the first one for cyclic tasks, mainly allocated to AOCS and thermal control., the second one dedicated to burst exchanges, mainly payload equipments.

For maximum safety and reliability, all equipments necessary to keep the satellite operating are redundant, while soft degradation is favored for the others. All computing and interface functions are redundant, as well as safe mode equipments, while performance equipments such as detectors, compressing chains, mass memory clusters, CMGs, star trackers are subject to soft degradation in case of failure.

Instrument

Optical Architecture

The optical solution chosen for the telescope is a Korsch type combination. The three mirrors arrangement gives an accurate combination, while keeping a compact design, and mirrors reasonably simple to build (curvature limits have been considered). An additional plane mirror has been used to enhance the compactness of the instrument. The overall dimensions are determined to keep a minimum distance between the primary and the secondary mirror, while keeping a reasonable distance between the secondary mirror and the tertiary mirror, so that the detectors are placed on the satellite side.

The imaging geometry optimization induces a primary mirror size of 650 mm diameter, which suits well to the detectors performance and the orbit characteristics.

Mechanical Architecture

The architecture chosen is organized around a central plane structure supporting the primary mirror, the tertiary mirror, the plane mirror, and a central cylinder that supports the secondary mirror. Attitude sensors (star tracker heads and gyroscope heads) are placed on this central instrument structure to improve the performance, as visible on Figure 1. A dedicated supporting truss structure ensures the instrument interface with respect to the bus. The detector thermal radiator has its own supporting structure.

The solution is based on Carbide material for the structure and on Zerodur material for the mirrors. This choice was made taking into account the peculiar properties of Carbide material : very low coefficient of thermal expansion, very low density, resulting in a light telescope, and a simple thermal control.
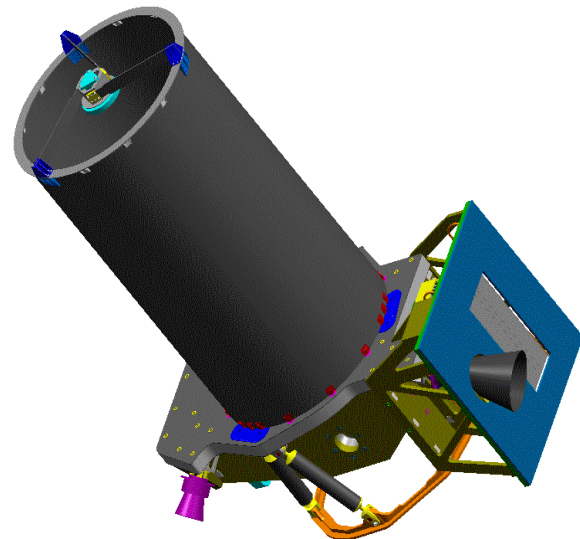
**Figure 3 : Instrument Configuration**

Mechanisms

The instrument includes a focus mechanism placed on the tertiary mirror. This position has been

determined to be an optimum between mechanism range and accuracy.

The instrument includes also an internal shutter to protect it from the Sun radiation in non-operational phases such as launch, attitude acquisition, or safe modes. This shutter is placed behind the primary mirror to protect only the tertiary mirror and the detection cavity. This solution avoids an external shutter that is generally heavy and complex. It has been shown that the front cylinder cavity can sustain a direct Sun illumination during safe mode transients without damage.

Detection

TDI detectors are used for panchromatic detection, with a maximum of 20 integration
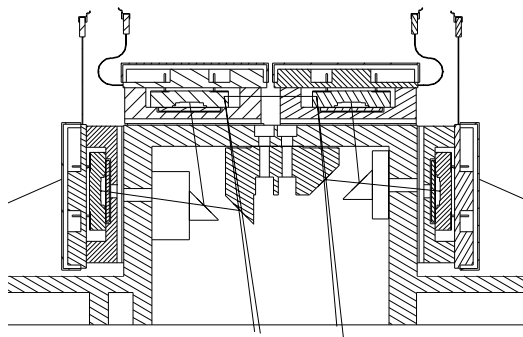
lines. Five detectors of 6000 pixels each are used; each pixel having a size of 13 μm. The multi-spectral detection is realized the same way, with 5 detectors of 6000 pixels each, 13 μm large. Each detector consists in a four lines assembly, enabling four colors imaging.

The focal plane is constituted by two symmetrical arrangements of those detectors. The beam splitter is made of a set of mirrors.

Figure 4 : Focal Plane Configuration

Each TDI detector outputs data at 58 Mpixel/s (or 290 Mpixel/s total), which corresponds to about 700 Mbit/s per detector (or 3.5 Gbits/s total in worst case). The multi-spectral detectors output at the same frequency their video data. Those video signals are converted in numerical data with individual chains at about 7 Mpixel/s,
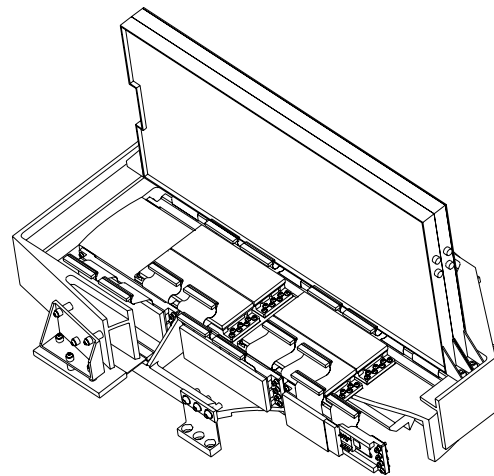
to keep a maximum performance, mainly in terms of signal to noise ratio.

The focal plane is physically coupled with the detection electronics to form the detection unit. The overall unit realizes the functions of detection and conversion of video signals in numerical data. The Printed Circuit Boards are placed just behind the focal plane as indicated in Figure 5. This highly integrated design allows a simplification of the data transmission between the focal plane and the detection electronics.

Figure 5 : Integrated Detection Unit

The detection unit has its own thermal control to ensure maximum performance. It is constituted mainly by a dedicated radiator, and by two heat pipes. The radiator is part of the instrument structure, and a hole is reserved on the coldest satellite panel for it.

Bus

Payload Data Handling and Transmission

The video data are output from the instrument at 4.5 Gbits/s total output rate, are compressed in the Payload Data Compression Unit. A wavelett algorithm is used, that enables the compression ratio to go up to 7, while in standard operation the ratio is 4. Image data are time tagged in this unit.

The compressed data are then memorized in the Solid State Mass Memory (SSMM). This memory has a Storage capacity of 600 Gbits. The maximum image data input rate is 1.5 Gbits/s. Auxiliary data

are collected on the 1553 bus interface. The output rate is nominally of 600 Mbits/s, on four individual channels of 150 Mbits/s each. The SSMM includes the function of ciphering for civilian purpose (AES). Data are packetized according to CCSDS standard. An auxiliary unit is used for military ciphering.

The data are then coded following a trellis-coded scheme in 8-PSK type modulators that include their own Solid State Power Amplifiers (SSPA). They are then multiplexed, and downlinked with an X-band antenna. The X-band antenna is a corrugated horn. It is mounted on a 2-axis gimbal mechanism to ensure image downlink during imaging and maneuvering sequences. This mechanism is not used during imaging to minimize the dynamic disturbances. A dedicated

antenna pointing algorithm is developed to orient the antenna during satellite maneuvers so that the ground station always stays in the antenna lobe. This way, the complete ground station visibility is used to downlink images.

Power

The power system uses a Li-ion battery, and triple junction solar cells. The 80 Amps-hour battery is directly connected to the power lines, and imposes its voltage. It is charged out of eclipse by the Gallium Arsenide (GaAs) cells of the 5 m² solar arrays. To ensure a balanced power budget over one day, the satellite points its arrays towards the Sun before and after every orbit imaging sequence, as indicated in Figure 6.
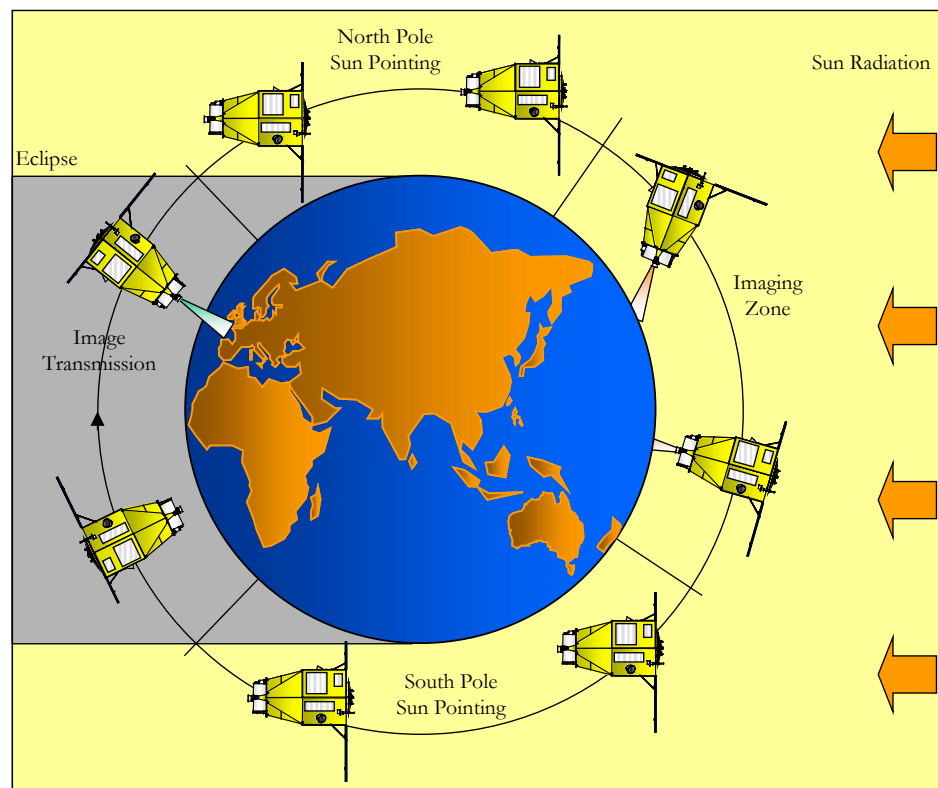
Figure 6 : Satellite Pointing along the Orbit

Li-ion batteries are well suited because of their high energy to mass ratio and the deep Depth of Discharge (DoD) possibly reached. They also

allow a very efficient charge sequence, which is highly indicated in the case of the fixed solar arrays solution. This way, the variable power coming on the

solar arrays is used optimally to re-charge the battery. The solar array size is therefore minimized. The use of triple junction GaAs cells also pushes towards smaller arrays, and a more compact configuration. The mission simulations performed show the DoD stays within very acceptable limits.

The power is managed by a Distribution and Regulation Unit (DRU). It uses Solid State Power Controllers (SSPC). It charges the battery according to a battery charge law determined in the central computer, and by switching the solar array section SSPCs. The power lines distributing the power to users are protected by individual SSPCs. Three types of SSPCs are used : the mono-stable that directly go to open circuit position in case of failure, the bi-stable that stay in their position in case of failure, and the rearmable that try to re-arm in case of failure. The mono-stable SSPCs are used for nominal equipments, that are not used in safe mode, the others are used for equipments that have to be used in safe mode.

Attitude and Orbit Determination

The autonomous orbit determination is performed by a Doris receiver. Doris is the Cnes navigation system, based on measurements between the satellite and dedicated ground stations at 400 MHz and 2 GHz. The receiver raw measurements are filtered inside the receiver by a high order navigator, based mainly on Earth gravity potential modeling, to reach an accuracy of about 1m. The receiver can be cold started in any satellite orientation in less than 1 orbit, which eases the satellite operations. It also gives the onboard time and the Pulse Per Second (PPS) necessary to synchronize the system.

The attitude determination is performed by a gyro-stellar system. The performance required imposes the use of three high accuracy star trackers, with separate optical heads. The star tracker is autonomous, and its accuracy is better than 2 arcsec as field of view error, and 10 arcsec (max) as noise. Only the axes perpendicular to the boresight axis are used to improve the accuracy.

In the same way, very accurate solid state gyroscopes are used to ensure high accuracy attitude determination while maneuvering. Fiber Optic Gyros (FOG) are chosen not only because of their low noise and very stable scale factor, but also because of their very low power dissipation at optical level. The gyroscope main performances are a scale factor stability of a few ppm, a random drift of 0.002 deg/h, and an angular random walk of 0.0002 deg/root-hour.
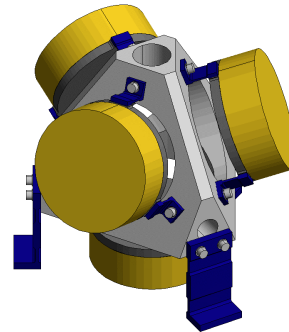
Figure 7 : Inertial Measurement Unit Optical Head Configuration

Both star trackers and inertial measurement unit have separated optical heads and electronic units. The optical heads are placed onto the instrument structure to minimize the thermal distortion with respect to the instrument line of sight. Moreover, the star trackers have a dedicated thermal control to minimize the couplings with the instrument, while the inertial measurement unit fully benefit from the instrument thermal environment. The resulting thermo-structural behavior is very stable, and remains in the same order of magnitude as the gyro-stellar noise.

Attitude Control

To reach the very demanding maneuvering requirements of the Pleiades satellite, Control Moment Gyros (CMGs) are mandatory. A cluster of four 15 Nms actuators are used.

Innovative guiding techniques are used to avoid the usual drawbacks of CMGs : instead of following a pre-defined attitude profile that always lock the cluster in singularities in some cases, a cluster re-orientation is realized taking into account the satellite trajectory and the cluster history to globally optimize the system. This new approach allows the use of the

complete angular momentum capacity envelope. The CMG cluster a priori guidance profile is determined in the angular momentum space by an adequate spanning. It has been shown that this reorientation strategy always avoids singularities, while ensuring a correct convergence towards the imaging dynamics. Moreover, this open-loop guidance law can be realized autonomously on analysis of the programming message. A dramatic simplification of GMG cluster management follows.

The satellite attitude control is performed by superposition on the hereabove mentioned cluster guidance law. The control bandwidth is at about 1 Hz, which ensures a tranquilization of less than 1 sec at the end of maneuvers.
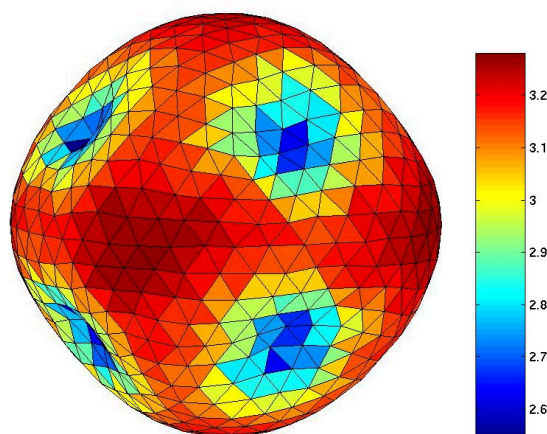
**Figure 8 : CMG Cluster Angular Momentum Capacity**

The CMG gimbal mechanism local control is realized at about 100 Hz sampling frequency, which ensures a complete frequency splitting, and avoids couplings with the AOCS. Furthermore, the ball bearing concept chosen for the CMG wheel is well adapted to the stiffness requirement induced by such a requirement.

The propulsion is grouped in a module that gathers all the related equipments : a tank of 50 l of useful hydrazine, four 1 N thrusters, fill and drain valves, plumbing, the local thermal control hardware. It is used only in the orbit control phases. This concept is directly derived from the Leostar development[1].

The non-operational modes are controlled as in the Leostar nominal design. The acquisition and Safe Hold mode (ASH) is based on the use of the magnetic based attitude control called 'B dot' law[1,2]. The orbit control modes are based on simple thruster off-modulation in open-loop, superposed to the nominal attitude control loop.

Performances Overview

The overall satellite weights 940 kg (max), and is compatible of Rockot, PSLV, and Soyuz launchers.
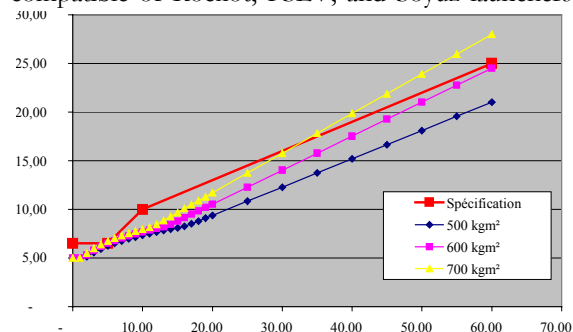
The corresponding inertias range from 600 kg.m² to 700 kg.m².

High agility performances are reached with the satellite compact design achieved, and the use of a cluster of 4 CMGs, even in case of inertia variation, as indicated in Figure 9. This agility coupled with the image storage and downlink capacities enhances the mission coverage to more than 50 imaging sequences per orbit, and more than 500 per day.

Figure 9 : Agility Performance (Maneuvering Time in sec wrt Angle in deg)

The mission achieves high accuracy images with performances indicated in Table 10.

Type	Parameter	Performance
Radiometric Image Quality	Pan MTF	0.09
	Pan SNR @ L2	90
	XS MTF	0.2
Geometric Image Quality	Image Centering	40 m (max)
	Stability during Integration	0.05 pixel (max)
	Absolute Image Location	20 m (max)

	Image Location on GCP	0.5 m (max)

Table 10 : Main Image Quality Performances

Perspectives

Pleiades phase C/D will begin in 2002 with a launch foreseen in 2006. The development and validation phases will be derived from the Leostar successful approach[2,3].

The use of new technologies, such as 13 μm TDI detectors, lightweight telescope structure, highly integrated detection unit, CMG, FOG, Li-ion battery will be validated through a technology pre-development phase already starting in phase B. Moreover, the use of new equipments derived from existing technologies will enhance the overall mission performance : the on-board computer, the star trackers, the payload data handling equipments are among those units.

Through those new concepts and technologies, Pleiades will be the next step of Europe in the High Resolution Earth observation area.

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